1991-09

A comparison of heavy lift launch vehicle options for the 1990's

Schreiber, Jonathan K.

Monterey, California. Naval Postgraduate School

20000828151

NAVAL POSTGRADUATE SCHOOL
Monterey, California

THESIS

A COMPARISON OF HEAVY LIFT LAUNCH VEHICLE OPTIONS FOR THE 1990'S

by

Jonathan K. Schreiber

September, 1991

Thesis Advisor: Dan C. Boger

Unclassified

Security Classification of this page

REPORT DOCUMENTATION PAGE

1a Report Security Classification Unclassified	1b Restrictive Markings
2a Security Classification Authority	3 Distribution Availability of Report Approved for public release: distribution is unlimited.
2b Declassification/Downgrading Schedule	
4 Performing Organization Report Number(s)	5 Monitoring Organization Report Number(s)
6a Name of Performing Organization Naval Postgraduate School — 6b Office Symbol *(If Applicable)* 39	7a Name of Monitoring Organization Naval Postgraduate School
6c Address *(city, state, and ZIP code)* Monterey, CA 93943-5000	7b Address *(city, state, and ZIP code)* Monterey, CA 93943-5000
8a Name of Funding/Sponsoring Organization — 8b Office Symbol *(If Applicable)*	9 Procurement Instrument Identification Number
8c Address *(city, state, and ZIP code)*	10 Source of Funding Numbers Program Element Number / Project No / Task No / Work Unit Accession No

11 Title *(Include Security Classification)* A COMPARISON OF HEAVY LIFT LAUNCH VEHICLE OPTIONS FOR THE 1990'S

12 Personal Author(s) SCHREIBER, JONATHAN K.

13a Type of Report	13b Time Covered	14 Date of Report *(year, month,day)*	15 Page Count
Master's Thesis	From To	1991 September	77

16 Supplementary Notation The views expressed in this thesis are those of the author and do not reflect the official policy or position of the Department of Defense or the U.S. Government.

17 Cosati Codes

Field	Group	Subgroup

18 Subject Terms *(continue on reverse if necessary and identify by block number)*
Saturn V, F-1, STME, Space Shuttle, Heavy Lift Launch Vehicle,Energia, Shuttle-C, ALS, Stafford Commission, Space Exploration Initiative, Advanced Launch System, Augustine Report

19 Abstract *(continue on reverse if necessary and identify by block number*

This thesis compares attributes of the heavy lift launch vehicle options that are available for the United States to use in support of the Space Exploration Initiative. The systems compared are the Advanced Launch System, Shuttle-C, Saturn V/F-1 and Energia. The F-1 development history is presented along with the proposed development of the ALS and Shuttle-C. Advantages and disadvantages of each of the systems are presented and possible trade-offs between them are discussed. The thesis shows that the option that is most cost effective and can be used to support the deployment of Space Station Freedom is that of developing a booster with a core engine like the F-1. In conclusion, a recommendation is given as to what the best option, with regard to both short and long term mission requirements, is for a new heavy lift launch vehicle for the United States.

20 Distribution/Availability of Abstract [X] unclassified/unlimited [ ] same as report [ ] DTIC users	21 Abstract Security Classification Unclassified	
22a Name of Responsible Individual Dan C. Boger	22b Telephone *(Include Area code)* (408) 646-2607	22c Office Symbol AS/Bo

DD FORM 1473, 84 MAR — 83 APR edition may be used until exhausted — security classification of this page

A Comparison of Heavy Lift Launch Vehicle Options for the 1990's

by

Jonathan K. Schreiber
Lieutenant Commander, United States Navy
B.S., University of Arizona, 1977

Submitted in partial fulfillment
of the requirements for the degree of

MASTER OF SCIENCE IN SYSTEMS TECHNOLOGY
(SPACE SYSTEMS OPERATIONS)

from the

NAVAL POSTGRADUATE SCHOOL
September 1991

Author: [redacted]

Jonathan K. Schreiber

Approved by: [redacted]

Dan C. Boger, Thesis Advisor

[redacted]

Michael G. Sovereign, Second Reader

[redacted]

Rudolf Panholzer, Chairman, Space Systems Academic Group

ABSTRACT

This thesis compares attributes of the heavy lift launch vehicle options that are available for the United States to use in support of the Space Exploration Initiative. The systems compared are the Advanced Launch System, Shuttle-C, Saturn V/F-1 and Energia. The F-1 development history is presented along with the proposed development of the ALS and Shuttle-C. Advantages and disadvantages of each of the systems are presented and possible trade-offs between them are discussed. The thesis shows that the option that is most cost effective and can be used to support the deployment of Space Station Freedom is that of developing a booster with a core engine like the F-1. In conclusion, a recommendation is given as to what the best option, with regard to both short and long term mission requirements, is for a new heavy lift launch vehicle for the United States.

TABLE OF CONTENTS

I. INTRODUCTION

On the twentieth anniversary of the first manned lunar landing, 20 July 1989, United States President George Bush put forth a challenge to the people of the United States:

> ... a long-range continuing commitment. First, for the coming decade, for the 1990s, Space Station Freedom, our critical next step in all our space endeavors. And next, for the next century, back to the Moon, back to the future, and this time, back to stay. And then a journey into tomorrow, a journey to another planet, a manned mission to Mars. Each mission should lay the groundwork for the next. [Ref. 1:p. 1-1]

This challenge was addressed by the National Aeronautics and Space Administration (NASA) Synthesis Group. Chaired by former astronaut Lt. Gen. Thomas P. Stafford, USAF (RET.), the Synthesis Group report, America at the Threshold, puts forth the Space Exploration Initiative which outlines four architectures developed in order to realize six specific "visions" which are meant to "guide and direct our space efforts" in order to meet the President's challenge with respect to the return to the Moon and the exploration of Mars. These four architectures are

1. Mars Exploration
2. Science Emphasis for the Moon and Mars
3. The Moon to Stay and Mars Exploration
4. Space Resource Utilization. [Ref. 2:p. 5]

These architectures are both broad in scope and aggressive in schedule. There will be, according to these architectures, a human mission to the lunar surface as early as 2003 and a human mission to the Martian surface as early as 2014 [Ref. 2:p. 5]. There are, of course, tremendous economic and technological differences between each of these four architectures but there are two factors that all four of them depend upon. The first is the restoration of a heavy lift launch capability and the second is the redevelopment of a nuclear propulsion capability [Ref. 2:p. 6].

The architectures outlined in the Space Exploration Initiative are focused more on the distant future whereas Space Station Freedom is garnering widespread public attention and funding right now. Although the exact details of the final configuration of Freedom are uncertain it is fairly certain that the final mass of Freedom in low earth orbit will be in the neighborhood of 1.5 million pounds, and it is scheduled to be manned and operating by 1998. Although some of the hardware for the Space Station can be launched by other, existing launch vehicles, the majority of the mass of Freedom is scheduled to be launched by the Space Transportation System (STS). With a maximum payload of approximately 40,000 pounds Freedom would require thirty eight fully dedicated launches of the Space Shuttle. At current launch rates for the Space Shuttle of perhaps six launches per year this would require more than six years just to get the hardware for Freedom into low earth orbit. This schedule would ignore the tremendous backlog of scientific and other payloads that resulted from the loss of Challenger on 28 January 1986. With that in mind Freedom would seem to be another part of the Presidential challenge that could benefit from the advent of another launch system that would be able to orbit the required mass in less time and, it is hoped, at less expense than that provided by the current United States inventory of launch vehicles. This sentiment is boldly stated in the executive summary of the Report of the Advisory Committee on the Future of the U. S. Space Program, commonly known as the Augustine report:

> We further conclude that NASA should proceed immediately to phase some of the burden being carried by the Space Shuttle to a new unmanned (but potentially man-rateable) launch vehicle. The new launch vehicle should offer increased payload capacity and be derivable wherever practicable from existing components to save time and cost. ...Such an evolving heavy lift launch system should be designed to produce substantial reductions in launch costs.... [Ref. 3:p. 7]

In January, 1959 President Eisenhower received a report from NASA that outlined a plan for a national space vehicle program. Authored by Milton Rosen, the report emphasized the lag in American rocket technology with respect to the Soviet Union and called for a new generation of large boosters. The report went on to say that the boosters that were in use at that time were designed for a limited mission and did not possess the design characteristics required by future needs of the National Space Program. [Ref. 4:p. 36] The National Space

Program at that time called for manned missions to the lunar surface and the establishment of a permanently manned space station [Ref. 4:p. 24]. Although this report was written more than thirty years ago its point is germane today. The conclusions of both the Stafford Commission and the Augustine Report also seem to point rather strongly to the fact that the United States again has a definite need for another launch system. The above quote from the Augustine report gives an initial feel for what capabilities a new launch system should provide, and President Bush's challenge defines the mission for which this new launch vehicle will be used. There seems to be very little difference between the mission needs of 1959 and 1991 nor does there seem to be much difference in how this mission need will be met.

Currently two systems that are undergoing program definition at this time have gained widespread support from NASA and the acquisition community. These are the Shuttle-C (Cargo) and the Advanced Launch System (ALS). The Shuttle-C is an unmanned derivative of the Space Shuttle and has the advantage of commonality with the current Space Shuttle but does little to improve the capability to lift mass to low earth orbit that the STS can provide. The Advanced Launch System is a joint U.S. Air Force and NASA program that is being studied and affords a moderate level of commonality with the Space Shuttle but will require an extensive development program with all the inherent risks therein.

Two other programs that have received less support from NASA and the acquisition community are the redevelopment of the Saturn V Heavy Lift Launch Vehicle (centered around the F-1 engine, this was the solution to the mission need of 1959), and the use of international launch systems, most notably the USSR's Energia.

The objective of this thesis is to assess the merits of requalification of a proven but dormant system versus the adaptation of an existing albeit other country system over the development of an entirely new system. By comparing the costs (relative to both treasure and time) of the Saturn V requalification or Energia adaptation to the costs of developing the Advanced Launch System or the Shuttle-C, it will be seen that the challenges of the return to the Moon to stay and manned exploration of Mars can best be answered by the use of systems that have already been developed.

II. THE MISSION

The ultimate goal of the Space Exploration Initiative is to re-establish the United States as a leader in space exploration. Towards this goal the broad mission requirements for the future have been outlined in the four architectures put forth in the report of the Stafford Commission [Ref. 2:p. 5]. Though these architectures are good for motivation and goal setting they do very little for designing and choosing the equipment that will ultimately be needed to realize these goals.

The specific mission requirements can be defined in any number of ways but the most widely used method is tied directly to the delivery of specific pounds of payload to specific orbits. From the pounds of payload to a certain orbit launch rates and schedules can be calculated for various mixtures of available launch systems.

According to NASA and Air Force documents the projected mass to low Earth orbit (LEO) in support of Space Station Freedom, the Space Exploration Initiative, Strategic Defense Initiative, and other Department of Defense and commercial missions is on the order of 5,000,000 pounds per year [Ref. 5:p. 22]. If the United States were to rely solely on the Space Transportation System (STS) this would require 125 launches per year assuming a 40,000 pound payload capacity for the Space Shuttle. Of course many of these launches can be performed by other systems such as the Titan IV, Atlas II, and Delta launch vehicles, none of which can carry greater than 6C,000 pounds to LEO. Regardless of the launch vehicle used 100 or more launches per year would require a successful launch every 3-4 days. The United States simply has not been able to support a launch rate this high. In order to still achieve the requirement for placing 5,000,000 pounds into LEO, several options could be pursued. The most widely agreed upon and seemingly efficient method of achieving this goal is to increase the payload capacity of available launch vehicles. This method makes sense both economically and statistically.

With regard to economics, costs for launch vehicles increase slowly with size and payload capacity and rise much more rapidly with increased reliability and complexity [Ref. 6:p. 4]. Therefore if the United States wishes to pursue a schedule of more than 100 launches per year the reliability and consequently the complexity of the launch vehicles will need to increase greatly from the present. Increased launch rates and increased reliability will cause costs to skyrocket. However, if the payload capacity of the United States launch vehicles were increased then fewer launches would be required. Reliability and complexity could be reduced if these new launch vehicles were unmanned. This would result in a lower cost solution to the problem of placing 5,000,000 pounds into LEO. Statistically, it makes intuitive sense that by reducing the number of launches then there will be a greater likelihood of mission success simply as a matter of not going into harms way as frequently.

With the mission defined as pounds to LEO, as opposed to the less quantitative definition put forth in the Stafford Commission report [Ref. 2:p. 5], then the design for the solution to the problem becomes; how can we best place 5,000,000 pounds into LEO?

III. THE SATURN V AND THE F-1 ENGINE

A. DEVELOPMENTAL HISTORY

At the height of the "Space Race" in August, 1958 the Advanced Research Projects Agency (ARPA) gave specific guidance and authorization of funds to the Army Ballistic Missile Agency (ABMA) for the development of a large space vehicle booster. This booster was to provide approximately 1,500,000 pounds of thrust. This new launch vehicle was known as the Juno V and was regarded by the director of ABMA, Dr. Wernher von Braun, as the realization of a dream and the beginning of Saturn. The Saturn designation was frequently used by von Braun and others at ABMA. [Ref. 4:p. 28]

Dr. von Braun and his team of engineers at Huntsville, Alabama had been doing extensive research on large boosters for a number of years. When the National Advisory Committee for Aeronautics (NACA) formed The Working Group on Vehicular Programs in January, 1958 it appointed von Braun as the chairman. When the final report from the working group was published in October, 1958 not only had NACA changed to NASA but von Braun was able to publicly present his ideas for a very large booster. This booster was to have a cluster of two to four engines of 1,500,000 pounds of thrust each, resulting in a total thrust of up to 6,000,000 pounds. [Ref. 4:p. 34]

Since NASA had been designated by President Eisenhower as the agency to conduct manned space flight programs it had considerable need for the type of booster that von Braun was proposing. NASA went on to develop their own large booster, the Nova. The Nova would incorporate the Juno V as an upper stage and was seen as the first launch vehicle capable of transporting a man to the lunar surface and returning him safely to earth. NASA focused the majority of its research and development attention towards Nova and Juno V and through the course of 1959 the Juno V program name was changed to Saturn. [Ref. 4:p. 37]

In order to fund the work on both Saturn and Nova it was decided that ABMA would take over the Saturn project. As it turned out, that put Saturn in the Department of Defense (DoD) which saw the Saturn program as too expensive and not of much military value and it was very nearly canceled in June 1959. Since NASA very much needed the Saturn program to support other projects, Milton Rosen and Richard Canright from ARPA spearheaded an effort to save Saturn. In October, 1959 that effort paid off and resulted in ABMA and Dr. von Braun being transferred to NASA. [Ref. 4:p. 39] By a presidential executive order issued on 15 March 1960 ABMA became Marshall Space Flight Center (MSFC) and Dr. von Braun subsequently became its first director [Ref. 4:p. 42].

This transfer would ultimately spell the end of Nova but would keep intact the brain trust that had the most expertise in the development of large space vehicle boosters and would make Saturn the preeminent booster program for the United States.

From 1958 through the time of the transfer of ABMA to NASA the Saturn program, as defined by ABMA, was to be designed for economy as well as power. The economic constraint was satisfied by exclusively using existing hardware, but there appeared to be no combination of existing hardware that was capable of providing the power necessary to meet the goals of the National Space Program. So something had to change. The first stage was deemed suitable and would not change. It would remain as a cluster of Kerosene (RP-1) and Liquid Oxygen (LOX) fueled engines providing 6,000,000 pounds of thrust at lift-off, but the upper stages were not suitable and would need to be changed.

During the time of the transfer of ABMA to NASA considerable research was done on using various combinations of modified upper stages of the Titan and Atlas vehicles for the upper stages of the Saturn. This proved to be a dead end in two respects. First the hybrid Atlas/Titan upper stages were not capable of providing the power that was desired and secondly the small diameter of these stages placed severe sizing constraints on the payload design. It was decided that the upper stages of the Saturn could not be realized from existing hardware and a whole new system would need to be developed. [Ref. 4:p. 44]

The answer to this design question came out of a committee chaired by Abe Silverstein, NASA's Director of Space Flight Development. The primary task of the Silverstein Committee was to select upper stage configurations for the Saturn [Ref. 4:p. 45]. The final recommendation that came out of the committee was that it would be best to use the controversial new high energy fuel of Liquid Hydrogen (LH_2) and LOX, referred to simply as Hydrogen fuels, as opposed to the conventional RP-1/LOX fuel, referred to as Hydrocarbon fuels. Many of the committee members, von Braun included, were skeptical of using untried engines and exotic fuels in a program initially designed to be somewhat low risk. With Silverstein as the primary advocate for the Hydrogen fuels the decision was made to base the upper stages of the Saturn on the engines of another new upper stage design called Centaur. Dr. von Braun and the rest of the members of the committee accepted this advocacy with the reasoning that there would be enough Centaur launches, prior to using these engines in the Saturn, to work out any problems and thereby reduce the risk to acceptable levels. [Ref. 4:p. 46] The Centaur turned out to be an unqualified success and is still in use as of this writing.

With the problems of the upper stages solved the focus of attention went to the definition of payloads and missions. After much discussion, debate and presidential direction a timetable was set for the ultimate mission of landing a man on the Moon and returning him safely to earth. With the mission fully defined there remained the problem of how to achieve the desired results. Depending on how a man was to be sent to the Moon different configurations of the launch vehicle emerged.

There were three primary methods discussed on how to get to the Moon. First was the direct ascent method whereby a Nova launch vehicle would boost a very large, and complex vehicle directly to the Moon. The Nova would consist of eight RP-1/LOX Rocketdyne F-1 engines in the first stage to provide nearly 11,000,000 pounds of thrust to boost the massive payload out of the Earth's atmosphere. The second stage would consist of two F-1s producing 2,700,000 pounds of thrust and would be used to adjust inclination and would establish the coarse trans-lunar injection. Then four hydrogen fueled Rocketdyne J-2 engines from the S-II upper stage (as agreed upon in the Silverstein committee) would provide 700,000 pounds of thrust for final alignment to the Moon. The fourth stage would consist of the S-IV with six Pratt and Whitney RL-115 Centaur engines that would provide 82,000 pounds of thrust for mid-course corrections and descent to the Lunar surface. The fifth stage would be two of the same Centaur engines as in the fourth stage and would provide 27,400 pounds of thrust and would be used for ascent from the lunar surface and return to the Earth. The objective of the direct ascent method was to liftoff from the Earth and land on the Moon without orbiting either planet. This meant that a very large and heavy vehicle would have to descend to the lunar surface and also liftoff from the surface. [Ref. 7:p. 158]

The second method was referred to as Lunar Orbit Rendezvous (LOR). This method called for a Saturn C-5 with five F-1 engines in the first stage, this was called the S-IC, providing 6,000,000 pounds of thrust, a second stage consisting of the S-II with five J-2 engines providing 910,000 pounds of thrust and a third stage consisting of the new S-IVB with one J-2 engine providing 182,000 pounds of thrust. [Ref. 7:p. 161] The main feature of this method was the weight savings realized by launching from the Earth and then, instead of landing directly on the Lunar surface, establishing an orbit around the Moon. From this orbit a smaller vehicle could be dispatched to the surface and an even smaller vehicle could return to the orbiting vehicle. This descent, landing and return with smaller vehicles was the keystone to the weight savings as compared to the requirements for the direct ascent method. [Ref. 7:p. 158]

The third method was called the Earth Orbit Rendezvous (EOR). This called for a series of Saturn C-5 launches placing the components for a Nova in low Earth orbit (LEO) and assembling them on orbit. After the parts had been assembled and the vehicle was manned, it would then proceed on a direct ascent to the Moon. [Ref. 7:p. 159]

Although it was generally accepted that all three methods were feasible a Langley engineer by the name of John Houbolt felt very strongly that the LOR was a simple cost effective scheme with high likelihood of success. Houbolt felt that the other two methods required boosters that were too large and lunar landers that were too complex. [Ref. 4:p. 64]

After much debate, most notably by Milton Rosen who favored the Nova direct ascent method [Ref. 4:p. 65], LOR was accepted by everyone concerned and the final configuration of the Saturn was solidified. The designation of the launch vehicle described in the explanation of the LOR method was subsequently changed from Saturn C-5 to Saturn V in early 1963. [Ref. 4:p. 60]

What this developmental history clearly points out is that during the development of the Saturn V the engineers involved were passionate about their mission but not to a fault. Each of the pivotal members of this history making evolution could clearly define what the solution to the problem would look like in their own eyes. But none were so adamant in their own design that a clearly stated and reasonable alternative was rejected. To wit, Dr. von Braun pursued his dream of a large booster using clustered hydrocarbon fueled engines. His dream had to be altered somewhat when Abe Silverstein made the case for using the exotic hydrogen engines to gain higher specific impulse at altitude. Milton Rosen salvaged the Saturn program by absorbing ABMA into NASA but had to demure to John Houbolt's more efficient method of fulfilling the mission and thus had to let Nova give way to Saturn V. Although not stated in this history but clearly remembered by those who lived through those times was the sense of urgency surrounding the "Space Race". President Kennedy motivated an entire nation by proclaiming on 25 May 1961 that;

> Now is the time to take longer strides, time for a great new American enterprise, time for this nation to take a clearly leading role in space achievement, which in many ways may hold the key to our future on Earth. I believe that this nation should commit itself to achieving the goal, before this decade is out, of landing a man on the Moon and returning him safely to the Earth. [Ref. 7:p. 154]

The magnitude and directness of this challenge resulted in the finest workmanship imaginable. It is possible that at the present time the level of motivation and dedication of resources that resulted in the Saturn V may not be achievable. It would be somewhat negligent to forego the dedicated efforts of thirty years ago and then attempt to duplicate the same effort in an entirely different motivational climate. As will be shown in the following section the efforts of thirty years ago resulted in a launch success rate, with respect to both schedule and cost, that has never been equalled.

B. THE F-1 ENGINE

Although the Saturn V was an integrated system consisting of three stages; the S-IC first stage, the S-II second stage, and the S-IVB third stage; most people associate the name Saturn V with only the first stage. Although this is inaccurate from a pedantic point of view the S-IC was indeed the impetus that ultimately lifted the massive payloads, that were required for the lunar missions, from the surface of the Earth. Without the S-IC the upper stages would not have mattered. Since the focus of this thesis is on the resolution of the lack of ability in the United States launch vehicle inventory to lift payloads in excess of 60,000 pounds into low Earth orbit it is on the first stage of the Saturn V that the remainder of this chapter will concentrate.

The central feature of the S-IC were the engines. The engines, designated F-1, were developed by Rocketdyne for an Air Force program in 1955. When NASA was formed in 1958 it absorbed several Air Force programs and the F-1 was to be one of the more important ones. The F-1 program came to NASA along with nearly all of the Air Force expertise and reports that had been done in previous years on these massive engines. After an in-house feasibility study was performed by NASA, Rocketdyne received a contract in 1959 to produce an engine with a thrust of 1,500,000 pounds. [Ref. 4:p. 105] The first production F-1 engines were delivered in October 1963 and the first S-1C cluster of five F-1 engines was tested at Marshall Space Flight Center in April 1965. The entire production run of 98 engines was completed in October 1969. [Ref. 8:p. 4]

The F-1 is considered to be a conventional engine using a combination of liquid fuel and a cryogenic oxidizer. The fuel used is kerosene (RP-1) and the oxidizer is liquid oxygen (LOX). The engine as delivered had a nominal thrust of

1,522,000 pounds at sea level and 1,748,200 pounds in vacuum and a specific impulse, which is the total impulse per unit weight of propellant [Ref. 9:p. 21], of 265.4 seconds at sea level and 304.1 seconds in vacuum. [Ref. 8:p. 2] The F-1 had a dry weight of 18,616 pounds [Ref. 8:p. 2]. Although the mission that would ultimately use the F-1 called for only one firing for a maximum of 166 seconds [Ref. 4:p. 408], they were qualified to 20 starts and 2250 seconds [Ref. 8:p. 2]. Ultimately 65 F-1s flew on thirteen missions and exhibited 100% reliability [Ref. 8:p. 2].

When the F-1 was first developed there existed no vehicle nor mission that would require such a huge engine [Ref. 4:p. 105]. Those shortcomings were short lived as the manned lunar missions took their final form and the final configuration of the Saturn V was confirmed on 10 January 1962 [Ref. 4:p.106]. As the final configuration of the Apollo missions were defined it became evident that the Saturn V was going to be required to place nearly 346,000 pounds into low Earth orbit in order to fulfill the manned lunar mission [Ref. 11:p. 2].

The F-1 engines were immense by any standard and the majority of the problems encountered with the development of these engines were connected with their sheer size [Ref. 4:p. 127]. It was found that simply scaling up from the 200,000 pound thrust H-1 engine, used in the Saturn I first stage, did not produce acceptable results. The major problems in the F-1 development were in critical areas such as the injector face, the turbo-pump, the thrust chamber and techniques for brazing the thousands of connections required in the engine. [Ref. 4:p. 109] But these problems were quickly and successfully overcome.

A testament to how well the F-1 engine was designed can be found in a test performed by Rocketdyne in 1972. The test was designed to prove that the Saturn V configuration would be acceptable to place Skylab into orbit. The only F-1 engines available for the Skylab mission had long since been put into storage. Therefore an engine that had been delivered to Marshall Space Flight Center in 1965, tested in 1966 and had since been stored was subjected to two extended firings in June 1972. The engine was then thoroughly inspected and analyzed; no abnormalities were found. [Ref. 4:p. 126] These tests paved the way for the Skylab mission to proceed so that on 14 May 1973 the last launch of the Saturn V boosted nearly 200,000 pounds of Skylab hardware into a 235 nautical mile circular orbit [Ref. 10:p. 5].

Although the Saturn V had performed its mission superbly it was abandoned for what was being viewed as the low-cost launch vehicle of the future. In Fiscal Year 1975 dollars it was projected that a reusable Space Transportation System would signal the end of the "brute force" period in launch vehicles for the United States. Payload delivery costs were expected to drop from the $800.00 - $1000.00 per pound range down to about $100.00 per pound or even less. With these tremendous savings payload flight requests had already dictated a launch rate of more than one per week. [Ref. 12:p. 7-14] With this new system in the spotlight the aging and expensive Saturn V was seen as a dinosaur and any further development or mission profiles that would benefit from this very highly successful launch system were canceled. The Saturn V was dead, long live the Shuttle.

C. FURTHER DEVELOPMENT OF THE F-1

The basic F-1 had a thrust of 1,522,000 pounds and a ten flight capability. With overhaul of the LOX turbopump, this lifetime could be extended to 20 flights. An extended life F-1 was tested at the same thrust rating with a 25 flight lifetime and a 50 flight lifetime with overhaul. A higher thrust derivative was also tested which had a thrust range from 1,570,000 to 1,800,000 pounds with the same lifetime as the extended life F-1. The biggest advantage of these engines were their reusability and throttling capability. [Ref. 8:p. 24] Reusability was achieved by strengthening many of the components such as the nozzle extension, the thrust chamber and the turbine exhaust manifold.

In the context of using the F-1 as an engine for launching payloads of the future it could be argued that the most important improvement to the basic F-1 would be the ability to throttle the thrust level. With the ability to throttle the thrust level there is a greater ability to provide for engine out capability. In the event that an engine fails during ascent it is the gimbal mechanism of the engine that was relied upon to provide for directional control of the spacecraft. The loss of thrust that results from the loss of an engine generally results in the payload not achieving the desired orbit. With throttling, the thrust of the other engines can be increased to compensate for the loss of at least one engine. That's one of the advantages of aircraft with more than one engine: the loss of an engine is not automatically catastrophic loss of the mission.

The throttling capability in the derivative F-1s was tested in two different modes. The first mode is step throttling where the thrust can be incrementally adjusted in set amounts. 194 tests were conducted on the F-1 with 50,000 to 70,000 pound thrust increments. The second mode is continuous throttling where the thrust is infinitely variable, on command, between certain limits. Seven tests were conducted with continuous throttling from 1,250,000 pounds of thrust to 1,840,000 pounds of thrust.[Ref. 8:p. 30]

As an example, assume a configuration of six F-1 engines. With each engine producing 1,250,000 pounds of thrust then the total thrust would be 8,100,000 pounds. If one engine were to fail the remaining five engines could be throttled to 1,620,000 pounds of thrust to compensate for the loss of the one engine. Although this is a very basic treatment of the problem of an engine loss on ascent, it shows that there is additional capability with the "brute force" F-1 than was used for the Apollo missions. It is interesting to note at this point that the only engine failure that was experienced during the Apollo program was experienced by the S-II second stage during the launch of Apollo 6. The S-IC stage worked as advertised but 4.5 minutes into the S-II second stage burn (five hydrogen-fueled J-2 engines), the number two engine shutdown followed shortly thereafter by the loss of the number three engine. The spacecraft still managed to gain orbital velocity and was not an entire loss. [Ref.4:p. 360] This is yet another testament to the robustness of the Saturn V launch vehicle.

The F-1 was not entirely without problems. The biggest problem encountered was a thing called "Pogo effect". This was not really a problem with the engines so much as it was a problem with the whole spacecraft. Pogo turned out to be a manifestation of the natural vibration frequency of the engines during their burn of 5.5 cycles per second. Near the end of their burn, at around two minutes, the natural frequency of the entire spacecraft was 5.25 cycles per second. This coupling of frequencies resulted in longitudinal oscillations of the spacecraft of around 5 cycles per second. This was not destructive but it was uncomfortable for the passengers and could cause unprogrammed stresses on the upper elements of the spacecraft. The solution to this problem was to "detune" the engine frequencies away from that of the spacecraft. [Ref. 4:p. 362] The point of this section is that any problems that were encountered with the Saturn V have long since been solved and are unlikely to haunt future launches.

D. F-1 ENGINE REFURBISHMENT

There can be little argument that the F-1 was a great engine in its day and that it performed its mission with aplomb. Although the performance of the F-1 was spectacular by any measure there is considerable ongoing debate as to whether or not the F-1 could be as successful today and in the future. Many of the arguments against the F-1 revolve around the difficulty and expense of re-establishing the production and test facilities that have long since been dismantled. There are also several questions as to who could supply the materials needed for such a large engine. The answer to these arguments is the subject of this section.

There are several F-1 engines scattered around the country. Rocketdyne Division of Rockwell International has identified thirteen that would be likely candidates for a refurbishing program [Ref. 8:p. 34]. There are five engines at Johnson Spaceflight Center in Houston, Texas that have been on display outdoors and uncovered. There are eight at the Michoud assembly facility in New Orleans, Louisiana. Of these eight engines, five have been on display outdoors and uncovered and three are stored in a checkout cell. The following refurbishment plan for these thirteen engines has been proposed by Rocketdyne and was briefed to the Stafford Commission in early 1991.

1. Full teardown and component evaluation of one engine. This would help determine the effects of long term storage and aid in the re-establishment of disassembly, assembly and checkout procedures.

2. Set the refurbishment plan timeline. This would be done by using existing overhaul specifications and the results of the teardown evaluation. There is experience for this type of procedure coming from Atlas, Delta and RS-27 overhauls.

3. Procure tooling and checkout equipment.

4. Activate a turbopump checkout facility.

5. Refurbish other engines for subsequent test and possible flight use.

The projected timeline to prepare and checkout one engine is twelve months, with subsequent engines undergoing minor refurbishment and delivered for testing at a rate of about one every two months thereafter. Minor refurbishment means no replacement of major parts and no turbopump green run. [Ref. 8:p. 36]

Projected rough order of magnitude costs in 1987 dollars for the F-1 engine refurbishment plan are shown in Figure 1. The non-recurring costs of engine disassembly and follow-on evaluation includes tooling, planning, drawing retrieval, procedure checkouts, etc. The per engine recurring costs that appear in the section for flight refurbishment assumes that no new hardware is required above and beyond that required in order to requalify an engine for flight. Under nearly ideal circumstances, this plan would be able to deliver a cluster of six F-1 engines at a cost of $15.8 million in roughly two years. [Ref. 8:p. 36]

Engine Disassembly and Evaluation

Non-recurring Costs	$1.1 million
Disassembly and Engineering Evaluation	$700,000
Total	$1.8million

Refurbishment for Flight

Non-recurring Costs	$2.0 million
Recurring Cost per Engine	$2.0 million

Figure 1. F-1 Engine Refurbishment Costs

This plan can be seen as a fairly inexpensive method of providing a limited number of engines for testing and possibly even short notice launch requirements. The problem of providing engines for sustained operations remains. In order to support a program of frequent launches the entire F-1 production infrastructure would need to be re-established. That will be the topic of the next section.

E. F-1 PRODUCTION RESTART

The problems associated with restarting a production facility for an old system are myriad but no more so than those associated with starting a new production facility for a new system. The primary elements of setting up a production facility are tooling, suppliers, materials, drawings and specifications, and test facilities.

The F-1 engine briefing that was given to the Stafford commission, as prepared by Mr. Paul Coffman of Rocketdyne Division of North American Rockwell, covers these topics and provides a preliminary timeline for the production of new F-1 engines.

The tooling for the F-1 engine was scrapped in 1976. The majority of the tooling has since become obsolete and is not compatible with current manufacturing techniques. With this in mind it can be seen that existing drawings would be of little use in a new F-1 engine facility; therefore, new drawings would need to be commissioned. Tooling that would be used in the manufacturing process numbers 3,840 items. There is a requirement for 2,736 items of tooling with respect to suppliers, and another 832 items for material handling. It is estimated that the cost to obtain this new tooling is $80,000,000.00 in 1987 dollars and would take 24 months.

With regard to tooling all is not bad news. There are some facilities being used for the Space Shuttle Main Engine (SSME) that could be used for the F-1. The thrust chamber assembly equipment and appurtenant facilities are available. The tube stacking room and the high-pressure flow area and injector brazing furnaces are also available. [Ref. 8:p. 38]

Sixty percent of the original F-1 suppliers are still in business and have expressed interest in participating in the rebirth of the F-1. For those suppliers that are no longer in business there are approved suppliers available for the remaining major parts such as the heat exchanger duct assembly, the liquid oxygen mating ring, the castings for the fuel pump housing, and the machining for the fuel pump housing. [Ref. 8:p. 40]

The estimated cost of reactivating all of the suppliers is $14,000,000.00 in 1987 dollars and would take 24 months. [Ref. 8:p. 40]

With regard to materials the news is even better. All of the materials that were use on the F-1 are still in use today with the exception of the material used to fabricate the 36 inch turbine manifold. That material was known as Rene 41. A replacement for Rene 41 has been identified and tested. The replacement is Hastelloy C-276 and was used to fabricate the 30 inch turbine manifold on the original F-1. To date two turbines have been built and tested and results indicate that there is no technical risk with Hastelloy C-276. [Ref. 8:p. 41]

The drawings and specifications that will be required for the reproduction of the F-1 will need complete review and update. The cost and time estimate for the complete overhaul of the engineering drawings and specifications is $1,000,000.00 in 1987 dollars and eight months. [Ref. 8:p. 42] During a phone conversation with Mr. Paul Coffman on 21 February, 1991 it was indicated that the estimate for the drawings was perhaps a bit optimistic and that five to ten million dollars was a more realistic estimate [Ref. 13].

The major stumbling block in the re-establishment of the F-1 engine production is the complete lack of any suitable test facilities. All of the test facilities that were used for the Apollo program have either been dismantled or have been converted to other uses. Again, the biggest problem associated with the F-1 test facilities is the sheer size of the engine and the tremendous thrust that it generates.

There already exists a location that tests the turbopump of the Atlas and the Delta launch vehicles. This test facility, known as Bravo II, could be modified to test the F-1 turbopump at a cost of $5,000,000.00 to $7,000,000.00 and could be ready in 24 months.

To test the engines for acceptance would require facilities at both the Jet Propulsion Laboratories test site and at Marshall Space Flight Center. The test site at the Jet Propulsion Laboratory would cost $60,000,000.00 and could be built in 36 months. A test site at Marshall Space Flight Center for an S-1C stage would cost $30,000,000.00 and would take 24 months to build.

These estimates were generated in the 1985-1987 time frame by personnel that had originally worked on the F-1. Parallels with the ongoing rework of the Atlas and Delta launchers were used as guidelines. The costs for the tooling were estimated using the top 20 cost drivers that were identified during the actual production run for the original F-1 in the 1960's. Although these costs and schedules may be found to be in error they were made by personnel that have actual experience on the system in question and, as much as possible, they were generated using actual data. [Ref. 13]

Figure 2 shows a summary of the estimated non-recurring costs and schedule for the re-establishment of an F-1 production facility.

Tooling	$80 million	24 months
Suppliers	$14 million	24 months
Materials		Already Exist
Drawings	$10 million	8 months
Test Facilities	$97 million	36 months

Figure 2. F-1 Production Costs and Schedule

With an estimated buy of 58 units the first F-1 production engine will cost $16,300,000.00 and the average cost per engine will be $12,700,000.00, in 1987 dollars [Ref. 8:p. 44]. These include only the recurring costs of the production of the F-1 and do not include development costs or any other non-recurring costs [Ref. 13]. Since the learning curve for the production of the F-1 is unknown, the cost of the Theoretical First Unit (TFU) will serve as the primary cost figure for comparison purposes.

F. F-1 SUMMARY

The F-1 engine has a proud history of success, and it had the advantage of being developed during a time when national pride was at an extremely high level. Truly the best and the brightest engineers, designers, managers and laborers were employed in an all out effort to produce a machine that would be capable of safely performing a mission that only a few years earlier was in the realm of science fiction.

The F-1 is a proven and reliable engine and even though it is not on the cutting edge of technology the F-1 would appear to offer some advantages that current and proposed systems lack. The primary advantages that the F-1 provides the United States are low risk and low initial investment. The lack of risk comes from the fact that the F-1 has been proven. The hydrocarbon engine, though lacking somewhat in specific impulse, is a simpler and less temperamental engine. It was these very reasons that Dr. von Braun opted for this type engine when he was faced with the problem of producing a high thrust, low cost launch system. Risk is also reduced by the availability of personnel that previously worked on the F-1. True, many of the key personnel have retired or have been more permanently removed from the industrial base but many are still available.

Lower investment costs are realized through the use of existing engines for test and evaluation. Additionally there will be little or no need for expenditures on research and development of technologies to support the F-1.

The most difficult task in re-establishing a production line for the F-1, according to Mr. Paul Coffman, will be the tooling and test facilities. Both of these problems will be present in any new system, also. [Ref. 13]

IV. OTHER LAUNCH SYSTEMS

A. THE ADVANCED LAUNCH SYSTEM

The Advanced Launch System (ALS) or National Launch System is a joint United States Air Force (USAF) and NASA project that hopes to fill the void in the United States heavy lift launch vehicle inventory. Although the project has not even made it to the demonstration and validation phase of the acquisition cycle a fairly well defined system has nonetheless emerged.

The ALS as currently envisioned is more a "family" of launch vehicles and the appurtenant testing, launch and support facilities versus a particular type of launch vehicle. The operational requirements for this family of vehicles is outlined in a briefing prepared by Mr. Ed Gabris of the NASA Office of Space Flight, Heavy Lift Launch Vehicles [Ref. 14]. The requirements in this briefing echo some of the same mission requirements as discussed in Chapter II above and some other additional requirements as well. Two requirements that remain the same are the pounds to LEO, 5,000,000 pounds, and the requirement to significantly reduce launch costs. The other requirements outlined in the briefing follow. ALS calls for a vehicle that is suitable for payloads that may vary from as little as 1,000 pounds to as much as 220,000 pounds. ALS reliability is to exceed 98 percent with a 30 day launch call up time. The ALS should have a 95 percent launch on schedule capability and a system-wide surge capacity of up to five launches in seven days. It must also be man rateable and be able to deliver the full spectrum of commercial and military payloads regardless of classification. There is also a requirement to deliver a payload to an operational orbit in the event of an engine failure during ascent. [Ref. 14:p. 9]

What this program turns out to be is a development scheme for an entirely new launch infrastructure and very nearly establishes a whole new paradigm for launch services in the United States. The wide range of payloads that are anticipated for this system make this program not only an addition to the STS but also the replacement for all vehicles other than the STS. In particular the Titan IV is scheduled to be replaced by the ALS in the 1999 timeframe and more than likely the same fate awaits the Delta.

This is not unlike the rationale that gave the United States the Space Shuttle. The overall objective in the ALS program is to produce a new vehicle and infrastructure that will provide the United States with the capability for high launch rates at low cost. Like the Space Shuttle program in the late 1960's and early 1970's the ALS program is also dependent on new technology to reduce flight costs and increase operational reliability [Ref. 14:p. 3]. Although there is considerable effort being made to use as much existing hardware as possible, the major elements of the ALS include the new Advanced Solid Rocket Motors (ASRM) and the new Space Transportation Main Engine (STME), both of which will be reusable [Ref. 14:p. 92]. The ASRMs will be refurbished in the same manner as the current STS solid rocket motors. The STMEs will be housed in propulsion and avionics pods that will be jettisoned after 180 seconds of use and be recovered and reused. [Ref. 15:p. 363]

Along with these main elements it will be necessary to construct new facilities to support the ALS. The additional expense for these new facilities are estimated to total 2.4 billion 1991 dollars [Ref. 14:p. 128]. Conversations with various experts on either side of the Saturn/ALS argument indicate that no matter what new launch system is developed there will be nearly the same expense incurred for facilities. Therefore for the purposes of this thesis these costs will be ignored. Also, since the ASRMs are being developed in support of the STS these costs will also be ignored.

Since the rocket engines themselves are considered to be the critical element of cost in any new launch system, [Ref. 14:p. 120], ignoring these other parameters seems reasonable. By basing the cost comparison on the primary cost driver associated with two different systems it is hoped that a clearer picture of the comparison between systems will emerge. With this premise established, the following section will examine the costs associated with the development of the STME.

B. THE SPACE TRANSPORTATION MAIN ENGINE

The Space Transportation Main Engine can be seen as a follow-on to the Space Shuttle Main Engine (SSME). The STME will be a gas generator, hydrogen engine (with liquid oxygen as the oxidizer and liquid hydrogen as the fuel). The STME is anticipated to have a thrust of 580,000 pounds in vacuum and a specific impulse of 429 seconds in vacuum. It will operate at a chamber pressure of 2250 psia and have a design reliability of 0.999. [Ref. 14:p. 123] As a comparison the SSME generates 488,000 pounds of thrust in vacuum, has a specific impulse of 453 seconds and operates at a chamber pressure of 2999 psia. [Ref. 15:p. 415] As can be seen, the STME will have increased thrust but will have reduced specific impulse and chamber pressure and will therefore operate at a lower temperature. It is hoped that design margins can be extended by using lower temperatures and pressures in the STME as compared to the SSME. It is also hoped that by using new manufacturing techniques, single point failures, i.e., failures that will cause complete mission failure, of the engine can be eliminated. [Ref. 14:p. 120]

In order to satisfy the requirement for payload flexibility the STME will be clustered in groups ranging from three engines to as many as 28 engines depending on the payload mass. Since these configurations are dependant more on the final design of the launch vehicle into which these engines will be put as opposed to the type of engine that will be used this topic will not be covered in any further detail.

C. PRODUCTION OF THE STME

Since the STME is essentially a brand new engine, there is quite an extensive development program associated therewith. First the particular technologies necessary to realize this engine need to be explored, prototyped and tested. The goal of these technologies is to develop an engine that is mass producible with low life cycle costs and high reliability. Currently these goals cannot be met by the hydrogen engines now in service or those used previously, to wit, the SSME, the J-2, and the RL-10 [Ref. 14:p. 30].

The key elements that need to be developed to make a low cost engine are a new liquid oxygen turbopump, a new liquid hydrogen turbopump, new turbopump manufacturing techniques, new combustion devices, new electromechanical propellant flow control systems, and a new engine controller [Ref. 14:pp. 31-44]. In the case of the turbopumps and the combustion devices the approach being taken in the ALS program is to derate pressures, temperatures and volumes from those used in the SSME for the same elements. In this way it is expected that there will be greater margins of safety built into these elements by not operating at material critical speeds, temperatures and pressures. There will be greater use of new manufacturing techniques in order to reduce the number of welds and parts required to manufacture each of these critical parts and in so doing reduce the number of stress points and, it is hoped, produce an inexpensive, mass producible element. Also, for these critical elements, the ALS program will place an emphasis on the use of conventional materials of greater weight versus exotic, lightweight materials as used in the SSME. This will help reduce the cost of raw materials and will provide greater margins of safety. The new engine controller and the propellant flow control systems will use solid state electronics and redundant logic to enhance reliability and will employ programmable circuitry to add flexibility. [Ref. 14:pp. 40-42]

It is anticipated that the STME program will start in fiscal year 1992 with the prototype development and achieve a first firing in 1995 and have an initial operational capability (IOC) in fiscal year 1999. The estimated cost to IOC is 1066.0 million 1991 dollars; this includes both recurring and non-recurring costs. With a 30 engine per year production rate it is anticipated that the theoretical first unit (TFU) will cost 11.46 million 1991 dollars. [Ref. 14:pp. 122-125]

D. THE SHUTTLE-C

The Shuttle-C is an unmanned cargo variant of the current Space Shuttle and is anticipated to be able to provide the United States with the heavy lift capability that will be required in the future [Ref. 16:p. 1]. The configuration that is currently being proposed will use the existing solid rocket motors, external fuel tank, and SSMEs from the current Space Shuttle. Payload capability of the Shuttle-C to the space station orbit will range from a minimum of 100,000 pounds for a two engine booster to 170,000 pounds for the three engine booster [Ref. 16:p. 1]. Possible follow-on improvements would be Advanced Solid Rocket Motors (ASRM) and STMEs [Ref. 17:p. 7], possibly housed in recoverable propulsion and avionics pods [Ref. 18:p. 9].

The most notable external difference from the current Shuttle and the Shuttle-C is the conspicuous lack of wings and vertical tail. The reason for this lack of conventional flying surfaces is that the Shuttle-C Cargo Element (SCE) will be an expendable payload delivery container [Ref. 16:p. 2]. The SCE will be a cargo hold fifteen feet wide and 82 feet long [Ref. 17:p. 3] (as compared with sixty feet long for the current Shuttle) attached to a boattail that will contain many of the same components that are in the boattail of the baseline Shuttle. Depending on the payload mass to be delivered the boattail on the SCE will contain either two or three SSMEs. Since these engines will be lost upon reentry, the SSMEs used on the Shuttle-C will be those engines that have been used on nine Space Shuttle flights and would be due for a major overhaul [Ref. 16:p. 2]. The loss of two or three $40,000,000.00 engines is justified by the reduced overall costs of placing payloads into orbit with an unmanned vehicle. This justification is based on the estimation that the Shuttle-C will be able to deliver its payloads to orbit for approximately $2000.00 per pound which is about half the cost of the current Space Shuttle [Ref. 19]. So the savings realized by the Shuttle-C would be twofold, i.e., lower launch costs and greater payload. For example, a 100,000 pound payload would only need one launch of the Shuttle-C at about $200,000,000.00 but would require two launches of the Space Shuttle at a cost of about $400,000,000.00. Therefore it would seem that the Shuttle-C should be able to deliver more payload at lower cost and at less risk than the Space Shuttle. These are costs that are projected for the initial design of the Shuttle-C. The follow-on version may result in greater savings but as of this writing there are no estimates available to support that claim.

Another advantage of the Shuttle-C is high commonality with the facilities and equipment that support the Space Shuttle. From a launch facility standpoint the Shuttle-C is identical to the baseline Shuttle and very few changes would need to be made to make flight practical [Ref. 18:p. 9]. This advantage would help expedite the transition from design to operations and it would also remove the 2.4 billion dollar investment that would be required for any other new launch vehicle.

It is anticipated that the first operational flights of the baseline Shuttle-C could occur three years after approval [Ref. 20:p. 8].

The Shuttle-C seems to be an attractive option to fill the void in the United States Heavy Lift Launch Vehicle inventory. Low development costs and quick operational capability are the strong suits of the Shuttle-C, the weak link in the Shuttle-C chain is the continued reliance on the SSME's with further upgrade to the STME's.

E. ENERGIA

The SL-17 Energia was introduced in 1987 as the main powerplant for the Soviet Space Shuttle, Buran, but is capable of launching other payloads as well in a side mounted cargo hold. The SL-17 looks something like the external tank of the Space Shuttle except that it has hydrogen engines at its base vice putting the engines in the orbiter as the U. S. chose. The most frequent configuration seen is, the large SL-17 with four, strap on K-1 boosters and either the Soviet Space Shuttle or the side mounted payload bay attached. The K-1 boosters are not at all like the boosters attached to the U. S. Space Shuttle. The K-1 boosters are attached to the core vehicle in linked pairs and are powered by hydrocarbon engines vice solid rocket motors. These boosters are jettisoned after approximately 170 seconds of flight. There may be provisions for their reuse. Energia has been used in two launches of the Soviet Space Shuttle, both of which were successful, and one launch of the side mounted payload carrier which failed to achieve orbit due to the failure of the orbit insertion module of the payload carrier. [Ref. 15:p. 431]

Energia has considerable flexibility with regard to the payload it is capable of delivering. This flexibility comes from the modular design of the booster. By varying the upper stages of the SL-17 and the number of strap-ons the payload delivery capability to LEO can vary from 336,000 pounds for four strap-ons to 408,000 pounds for six strap-ons. It is also capable of delivering 43,000 pounds to geostationary orbit, 77,000 pounds to the Moon, or 67,000 pounds to Mars/Venus. [Ref. 15:p. 431]

Since the fall of the Warsaw Pact and the opportunities that became available through glasnost and perestroika, Glavcosmos, which is the Soviet Union's equivalent of NASA, has offered their launch vehicles commercially. The Soviets have successfully sold their launch services to India, France and Germany. They have also sold a ride to the MIR space station to a Japanese journalist for $12,000,000.00. In the United States, a Houston, Texas based, privately owned company known as Space Commerce Corporation (SCC) has the sole U. S. marketing rights for these services. [Ref. 21:p. 91] According to the executive vice president and chief operating officer of SCC, Mr. William B. Wirin, the pricing structure for these services is highly negotiable. A rule of thumb that Mr. Wirin uses to estimate costs for launch services provided by the Energia is $300,000,000.00 to $360,000,000.00 for a 220,000 pound payload to LEO. [Ref. 22] Although these are rough order of magnitude costs but they translate to $1400.00 to $1600.00 per pound of payload to LEO, which is very competitive in today's market.

There are, of course, very serious political and economic implications associated with the use of an international launch system (especially those of the Soviet Union) but those issues are beyond the scope of this thesis. But if the mission is defined as a certain number of pounds of payload to a specific orbit in a limited timeframe then, from solely a mission view point, the option of using the Soviet Union's launch services may well be a most pragmatic solution to the problem of delivering payloads to space.

V. COMPARISONS

A. INTRODUCTION

In order to compare the various launch systems herein described it is necessary that all of the systems be viewed in the same context. For the purposes of this thesis that context will be the ability of a particular launch system to lift the requisite mass for Freedom to be manned and operating by the end of calendar year 1999 and to do so in a cost effective manner.

This date is chosen because President Bush's challenge of 20 July 1989 stated that the establishment of Freedom in the 90's is our next critical step in the exploration of outer space [Ref. 1:p. 1-1]. As the end of 1991 approaches that leaves precious little time for systems to be developed, tested and certified. Therefore the system chosen must be fairly low risk in that there is not much time to wait for developing technologies to mature to the point of useful application. In other words, the United States needs to utilize a system that is practical in terms of the technology available today.

It will be clearly evident to all concerned whether or not the constraint of timeliness is ultimately satisfied but knowing whether or not the constraint of cost effectiveness is satisfied will be a bit more difficult to discern. It is not necessarily true that the system with the lowest price tag will be the most cost effective. Although low price will be a major contributor to cost effectiveness other attributes will need to be considered as well.

These other attributes have been defined by the National Space Council. On 24 July 1991 Vice-President Quayle, chairman of the National Space Council, announced the National Space Launch Strategy. In this strategy, new space launch systems are expected to improve the national launch capability by reducing operating costs and by improving reliability, responsiveness and mission performance. The new launch system, including its supporting infrastructure, will be designed to support medium to heavy payload requirements and facilitate evolutionary change as requirements change. The new launch system is expected to be unmanned in the early operational environment but must be man-rateable in the future. [Ref. 23:p. 3]

With regard to reliability, the current rate of all U. S. launch vehicles is 92 percent. [Ref. 22] Therefore, without any other direction, it will be assumed that the goal for reliability of any new launch system is the same as that set forth in the Advanced Launch System operational requirements of greater than 98 percent for a particular system [Ref. 14:p. 9]. These same requirements can be used to set goals for responsiveness and mission performance in the new national launch system as being required to meet a 95 percent launch on schedule rate with a 30 day launch call-up time. The new launch system must be able to change out payloads in a maximum of five days and be able to support a system wide surge capacity of seven launches in five days. [Ref. 14:p. 9]

With regard to reduced operational costs considerable direction is derived from Public Law 100-180 in the Department of Defense Authorization Act 1988/1989, Section 256 (101 Statute 1066). This law requires that the Advanced Launch System will be required to lower recurring launch costs per pound by a factor of ten as compared to current expendable launch vehicle costs. For the purposes of this law, current launch costs are considered to be $3000.00 per pound to LEO in 1987 dollars. [Ref. 24:p 71] This would mean that a new launch system would be required to reduce recurring launch costs to $300.00 per pound to LEO (although it is unclear as to whether or not this law will apply only to the Advanced Launch System or to any new launch system regardless of its name). It is interesting to note that this is very nearly the cost per pound goal of the Space Shuttle in 1973 [Ref. 12:p. 7-14].

With the aforementioned constraints and selected goals as reference points for comparison, the following sections of this chapter will explore the advantages and disadvantages of each of the launch systems herein discussed.

B. PROPULSION COMPARISON

What has been discussed so far with respect to the propulsion systems that will be used on a new launch vehicle has been a hydrocarbon fueled engine, the F-1, and two hydrogen fueled engines, the STME and the SSME. It is assumed that the oxidizer of choice will be liquid oxygen in all cases.

When comparing liquid rocket fuels, several factors need to be considered before an intelligent choice between types can be made. The major factors that need to be considered are economic factors, performance characteristics, and physical hazards.

1. Economic Factors and Physical Hazards of Fuels

It is desirable to use a fuel that is economical, i.e., it is available in large quantities and at low cost [Ref. 9:p. 168]. In the case of liquid hydrogen (LH_2) and RP-1 it is clear that RP-1 is the least expensive of the two by a factor of five. In 1991, for quantities in the range that are expected for a heavy lift launch vehicle, the price for a pound of LH_2 was $2.00 and the price of RP-1 was $0.40 [Ref. 25].

The production process for a fuel should be simple and should not require special equipment or exotic raw materials [Ref. 9:p. 168]. RP-1 is a specifically refined petroleum product very much like kerosene and is easily produced in large quantities with very minimal dangers [Ref. 9:p. 179]. LH_2 on the other hand is the coldest fuel of choice. This low temperature causes problems with regard to both economics and physical hazards.

The low temperature of LH_2 causes problems with tanking and piping because most metals lose their strength at such low temperatures. Therefore special materials need to be used for the containment of LH_2 as opposed to the simple materials used for the containment of RP-1. The pipes and tanks must also be well insulated in order to avoid the formation of solid or liquid air or ice on these structures. Also, due to the low temperature, all common liquids and gases solidify in LH_2. These solid contaminants can cause plugging of orifices and valves. Therefore extreme care must be taken to purge all lines and tanks of air and moisture by pulling a vacuum or flushing with helium prior to introduction of the LH_2. Another disadvantage of LH_2 is that when mixed with solid oxygen or solid air the mixture is highly explosive. [Ref. 9:p. 180]

Small molecular size is another problem with LH_2. This small size requires special attention be paid to seals and welds to avoid leaks through fittings and assemblies [Ref. 25]. These problems with LH_2 have obviously been overcome through the years but remain a source of added expense and risk when compared to the use of RP-1.

2. Fuel Performance

Fuel performance can be measured a number of ways but the most used measure and that used in this thesis is specific impulse. Specific impulse is the total impulse, or thrust force, per unit weight of propellant [Ref. 9:p. 21]. More simply, specific impulse is a measure of how efficiently a fuel can move itself. LH_2 has a specific impulse of around 400 seconds and RP-1 has a specific impulse of around 300 seconds. This means that a pound of LH_2 can provide one third more total force than a pound of RP-1. But this does not necessarily mean that LH_2 is better than RP-1 at moving rockets from the Earth to LEO. Although the specifics of fuel performance is beyond the scope of this thesis it is important to understand how this difference in specific impulse will affect the cost of a launch vehicle. There are three important factors that directly affect launch vehicle costs as a result of the type of fuel chosen. The first is that LH_2 is a much less dense fuel than RP-1, therefore it will occupy more volume given an equal mass [Ref. 9:p. 180]. Secondly RP-1 engines have less mass for a given thrust requirement than LH_2 engines. And third, the gross mass of a launch vehicle increases with velocity more rapidly with RP-1 than with LH_2 [Ref. 26:p. 93].

The first two factors are simply a result of the physical properties of the fuels and how they are used. The third point is germane in that at low velocities, 8 kilometers per second and less, RP-1 fueled vehicles have a lower dry vehicle mass per fuel mass ratio than LH_2 vehicles. For reference, 8 kilometers per second is approximately 5 statute miles per second and is sufficient velocity for orbits above 100 miles. At velocities greater than 8 kilometers per second LH_2 provides a lower dry vehicle mass per fuel mass ratio than RP-1. What this means is that booster stages will be smaller and lighter if powered by RP-1 as opposed to LH_2. Therefore from launch to orbital injection, RP-1 fueled vehicles lead to lower dry masses than LH_2 fueled vehicles. [Ref. 26:p. 93] It also means that LH_2 will provide lower dry mass per fuel mass ratios than RP-1 once the vehicle is in orbit. Since vehicle production costs tend to vary as a direct function of dry weight then minimizing dry weight would be an important consideration when trying to minimize launch vehicle costs [Ref. 27:p. 248]. This is one reason that Dr. von Braun found it necessary to support the recommendations of the Silverstein committee report in 1959 [Ref. 4:p. 46].

Therefore fuel performance is not just tied to the ability of the vehicle to lift mass but is also a major consideration in the cost of the launch vehicle. RP-1 has the capability of providing the most efficient boost from the ground to orbital velocity and LH_2 is most efficient at providing the velocity changes required once the vehicle is on orbit.

3. Fuel Performance Summary

It is evident that RP-1 is more economical, less dangerous, and provides for a more efficient velocity change from the launch pad to LEO as compared to LH_2.

RP-1 is one fifth the price of LH_2 and can be handled by conventional means. RP-1 results in more efficient velocity changes from the launch pad to orbital velocities and will reduce the cost of the booster by reducing both its weight and its size. But it is also more efficient to use LH_2 for on-orbit velocity changes than RP-1. What this leads to is a two-fuel launch architecture in order to maximize efficiency and minimize costs.

The next section will, very briefly, discuss the major trade-offs that are pertinent to the argument between single fuel launch architectures, hydrogen booster and upper stages, and two-fuel launch architectures, hydrocarbon booster and hydrogen upper stages.

C. SINGLE FUEL VERSUS TWO FUEL ARCHITECTURES

Telephone conversations with Mr. Tom Irby, Air Force Systems Command, Space Systems Division, Mr. Billy Shelton, NASA, Marshall Space Flight Center, and Mr. Harry Cikanek, Technical Manager, STME Phase B Studies, Marshall Space Flight Center have all indicated that there are two major problems with adopting a two-fuel launch architecture. First there would need to be two engines, and second there would need to be two sets of ground support equipment for the storage and transfer of two types of fuels. [Refs. 28, 29, 30]

With regard to the problem of developing and maintaining the facilities to support two engines, Mr. Harry Cikanek believes that economies of scale would be lost. The reasoning for this is that if the same engine can be used for the booster and the upper stages, the STME in this case, then the production rate can be higher. With a higher production rate then there will be some cost savings as a result of the projected 94% production rate curve for the STME. These projected savings are expected to offset the added expense of using a less efficient engine for the boost phase [Ref. 30]. Quantitative data supporting or refuting this argument is not available.

In the case of the need to build more ground support equipment, i.e., facilities for the storage and distribution of two fuels, Mr. Irby and Mr. Shelton believe that this would add greatly to the non-recurring costs of a new launch system. These added costs coupled with the added costs of the development of another engine would drive the initial investment cost of a new launch system too high. [Refs. 28, 29] Again there is no available quantitative data to support this argument. But there is a historical parallel.

In the case of both the Atlas and the Delta Medium Launch Vehicles the booster uses RP-1 and the upper stages use LH_2 [Ref. 15:pp. 413-414]. Both of these systems have been in operation since the 1960's and the supporting infrastructure is still intact. Therefore it would seem unlikely that there would need to be terribly great added expense to adapt these facilities to the use of a new launch system [Ref. 13]. Again, there is no quantitative data to support this argument.

D. COST COMPARISONS

1. F-1 Versus STME

a. *Dollar Costs*

In Chapter III above it was stated that the Theoretical First Unit (TFU) cost for the F-1 is $16,300,000.00 in 1987 dollars [Ref. 8:p. 44]. Chapter IV showed the cost of the STME TFU to be $11,460,000.00 in 1991 dollars [Ref. 14:pp. 122-125].

In order to accurately compare the costs of the STME and the F-1 it will be necessary to adjust the F-1 engine TFU cost for time. By using Department of Defense deflators for aircraft procurement it is possible to bring the 1987 cost of the F-1 TFU up to the level of 1991 dollars.

The use of the deflator for aircraft procurement was deemed appropriate for two reasons. First, a similar deflator for rocket engines could not be found. Second, many manufacturing techniques used for the production of modern aircraft are used in the production of rocket engines.

According to Data Search Associates of Fountain Valley, California, costs that are estimated in 1987 dollars can be adjusted to 1991 dollars by dividing the 1987 costs by 0.856 [Ref. 31:p. D-1]. Equation (1) shows the result for the TFU of the F-1.

$$\frac{\$16,300,000}{0.856} = \$19,042,056.07 \tag{1}$$

Since the cost of the first unit of the STME in 1991 dollars is $11,460,000.00 and the cost of the first unit of the F-1 would be slightly more than $19,000,000.00 in 1991. This being the case it would seem that for a given production run and learning curve then the cost of the STME procurement will be less than that for the F-1.

But this may be an unfair comparison since the F-1 provides nearly three times the thrust of the STME at only 1.7 times the cost of the TFU of the STME. So if these costs are adjusted further for the propulsive capability that each engine is able to provide, a different conclusion can be reached. An adjustment of this sort seems appropriate because the primary function of a rocket engine is to propel mass. So it would seem that a better measure of cost effectiveness for a rocket engine is how much thrust is provided per dollar as opposed to the total cost for a particular engine. The adjustment that follows is conservative in that each engine will be adjusted for the thrust capability in its most efficient regime. For the F-1, sea level thrust will be used since hydrocarbon engines are best from launch to orbital velocity and vacuum thrust will be used for the STME for a similar reason. Vacuum thrust is always a larger number than sea level thrust. Also, the basic F-1 thrust of 1,522,000 pounds will be used versus the derivative thrust of 1,870,000 pounds. Equation (2) shows the propulsive adjustment for the STME and Equation (3) shows the adjustment for the F-1.

$$\frac{\$11,460,000}{580,000\,pounds} = 19.76\frac{\$}{pounds} \tag{2}$$

$$\frac{\$19,000,000}{1,522,000\,pounds} = 12.48\frac{\$}{pounds} \tag{3}$$

What these two adjustments show is that the F-1 provides its thrust at 63% of the cost of the thrust provided by the STME.

To summarize, although the cost of the TFU for the F-1 is greater than that for the STME, the F-1 is able to provide a given amount of thrust at less cost than the STME. The implication is that if a given thrust to weight ratio is required at lift-off for safety reasons then the F-1 will be able to provide this thrust for 63% of the cost that the STME can provide the same thrust. Additionally since the dry mass of a STME powered vehicle will be higher than the dry mass of an F-1 powered vehicle then the STME powered vehicle will be required to provide even more thrust than the F-1 powered vehicle in order to maintain the same margin of safety.

Another way to compare the cost effectiveness of the STME to that of the F-1 would be to evaluate the results over the course of a mission model. In this way the costs for the procurement of a number of engines that will perform a specific mission can be compared. For the purpose of this particular comparison the mission will not be limited to just that of deploying Freedom but the more broad scale mission as described in Chapter II above. Chapter II stated that the overall mission requirements were on the order of 5,000,000 pounds of payload to LEO per year. The last heavy lift launch vehicle, in terms of delivered payload, for the U.S. was the Saturn V used during the Apollo missions. This will be the model of the vehicle that will be used in this comparison. Since this is a first order approximation there is no attempt to adjust vehicle dry mass when the STME is substituted for the F-1. Also, in order to remove the complications of inflation and discounting over time, a one year launch cycle will be assumed. Therefore the model will compare the costs, for the engines only, to lift 5,000,000 pounds of payload from the launch pad to first stage burnout with a Saturn V that is powered by either F-1 engines or the STME.

The model will use the approximate vehicle mass for the Apollo/Saturn mission SA-509. In this mission the vehicle liftoff weight was approximately 6,423,754 pounds. Of this, 6,311,478 pounds were consumed or jettisoned prior to the payload being established in LEO. This left a payload in LEO of 112,276 pounds. [Ref. 32:p. 1-6] To simplify the arithmetic, a 100,000 pound payload will be assumed. At liftoff, the five F-1 engines were providing approximately 7,650,000 pounds of thrust [Ref. 32:p. 1-6]. This provided a thrust to weight ratio of about 1.2 to 1. If the STMEs were used instead of the F-1s then 13 engines would be required in order to maintain the same thrust to weight ratio. Since this model vehicle is delivering 100,000 pounds of payload to LEO then 50 launches would be required to complete the stated mission. That would require 250 F-1s or 650 STMEs. This assumes the vacuum thrust level for the STME.

Although it is highly unlikely that this many engines could reasonably be expected to be produced in one year this will be assumed for this model. In order to estimate the total cost of the procurement for either the F-1s or the STMEs a learning curve needs to be applied to the production run. The learning curve has the effect of lowering the cost as production quantities double. The learning curve that will be used in this model will be that which was u[illegible] to estimate the cost of the TFU for the STME. Mr. Cikanek said that the costs for the STME were estimated using a 90%

learning curve up until about June of 1991. Since then the learning curve has changed but a final figure is not available. [Ref. 30] For the purposes of this model the same learning curve will be applied to the production of the STME and the F-1. This seems reasonable in that the processes required for the production of either engine will be about the same. Equation (4) is the relation that will be used to estimate the total cost of a production run of 250 F-1s and 650 STMEs:

$$TC_{CUM.AVE.} = AX^{B+1} , \qquad (4)$$

where $TC_{cum.\ ave.}$ is the total cost of a production run using the cumulative average costs for the item being produced (this could be done with unit costs but the result will not change appreciably), A is the cost of the first unit or TFU, X is the number of engines produced, and B is Log_{10} learning curve divided by Log_{10} 2. [Ref. 33] Equations (5) and (6) show the result for the F-1s and the STMEs, respectively.

$$TC_{F-1} = (\$16,300,000)(250)^{.848} = \$1,760,519,062.00 \qquad (5)$$

$$TC_{STME} = (\$11,460,000)(650)^{.848} = \$2,783,140,583.00 \qquad (6)$$

The results of Equations (5) and (6) clearly show that there is more than a billion dollars difference between the total cost~ for the STME and the F-1 in this mission model.

A factor that is not taken into account in this model is the effect of production rate. Production rate acts similar to learning curve with respect to total costs. In the case of the F-1 the TFU was predicated on a production run of 58 engines [Ref. 8:p. 44], and the TFU for the STME used a production run of 30 engines [Ref. 14:p. 123]. The production rate factor for the STME was estimated by Mr. Cikanek to be 94% [Ref. 30]. Since this model is a first order approximation and the coupling of production rate with learning curve is more of a second order effect this coupling will not be done here [Ref. 33]. But by lowering the cost of the TFU for the STME to a point where the total cost of its production run equals that of the F-1 an approximation for the effect of production rate can be seen. In this case the TFU for the STME would need to be lowered to the vicinity of $7,250,000.00. Which would be a 63% reduction in the quoted cost of $11.46 million for the STME TFU. It is not reasonable to assume that a production rate of 94% could result in this amount of reduction in the total cost for the STME production run.

In summary, the F-1 can provide its thrust at less cost than the STME, 19.76 dollars per pound for the STME and

12.48 dollars per pound for the F-1. Also, because of the higher thrust level provided by the F-1 compared to the STME, fewer engines will need to be built. In the case of the model presented above this results in more than a billion dollars reduction in engine production costs.

b. Time Costs

As previously stated, it will be easy to determine if the time constraint imposed on a new system is satisfied. Either the new system will be able to provide services in the allotted time or it will not. In the case of the STME the projected Initial Operational Capability (IOC) is in calendar year 2001 [Ref. 14:p. 121]. This will obviously not satisfy the time constraint of providing heavy lift capability to support the deployment of Freedom prior to the end of calendar year 1999. On the other hand, production F-1 engines are projected to be operational approximately 4.5 years after approval of funds and refurbished F-1 engines may be available 2.5 years after approval of funds. [Ref. 8:p. 43]

The F-1 may not be able to satisfy the time constraint of providing services for the deployment of Freedom but it certainly has a better chance of doing so than the STME.

2. Shuttle-C versus Energia

In the context of this treatment this comparison seems appropriate in that both of these systems can provide roughly the same payload mass to LEO. Also the development costs involved are mainly focused on the vehicle and the supporting infrastructure and do not include the propulsion system. In the case of Shuttle-C the payload carrier needs to be developed and proven and in the case of Energia mating of U. S. payloads to a Soviet launcher may require extensive work.

It may or may not be a valid assumption that mating U. S. payloads to a Soviet booster and the logistical requirements needed to deliver these payloads to the Soviet launch sites will not cost any more nor take any more time than the development of the payload carrier for the Shuttle-C. But this is the assumption that is made for this section. That being the case, then the comparison between Shuttle-C and Energia is moot. Shuttle-C and Energia appear equally likely to be able to perform the mission of deploying Freedom by the end of the 1990's and their dollar costs for a pound of payload to LEO are comparable, also.

The major comparison between Shuttle-C and Energia is a topic beyond the scope of this treatment. Shuttle-C is planned to use SSMEs that are ready for major overhaul. The question arises, can the SSMEs provide the reliability required for the mission? The main concern in this context is, of the Shuttle delays due to engine malfunction how many of these delays would not have occurred if the Shuttle was unmanned? Research into this question yielded no information.

Perhaps this is a too subjective question to ask but it might be a primary schedule driver if Shuttle-C is developed.

The reliability of Energia is 100% but this is derived from a very limited sampling and the source of the information is not necessarily the most reliable. [Ref. 15:p. 431].

The other main topic for comparing the Shuttle-C and Energia are the politico-economic implications of using a Soviet launch vehicle. This would be a good topic for future research but is beyond the scope of this thesis.

VI. CONCLUSIONS

The F-1 engine was developed in a motivational and economic climate that may be impossible to reproduce in the near future in the United States. The development of the F-1 and the vehicle that it powered was in response to a challenge not unlike that which is facing the United States today. Support for the F-1 draws it strength from the exhibited reliability during the Apollo program and the follow-on developments that improved on its inherent capabilities, most notably the ability to modulate its thrust. Also the F-1 is a more efficient propulsion system than the STME in the low velocity regime. This efficiency will help reduce the overall cost of a new launch system by reducing both the cost of the fuel used and the cost of the vehicle that is built. This is a strength derived from the immutable laws of physics. However, the predicted economic efficiencies that might be gained from economies of scale for the production of the STME are based on the laws of economics, which are more difficult to use for accurate predictions.

A new version of the F-1 engine can be ready for operational use in roughly half the time of the STME and might therefore be able to help deploy Freedom and begin the Lunar and Martian missions on schedule.

The option of developing the Shuttle-C is attractive except for the propulsion system that it will use. More research needs to be done to see if the Shuttle-C would be capable of performing within the schedule constraints required for deploying Freedom by the end of 1999.

Energia is an attractive option except for the uncertainties involved in knowing its true reliability and performance. This says nothing of the political uncertainties of using Energia. This option might benefit greatly from continued research.

The goal of the STS was to lower the cost of placing payloads into LEO to a figure in the low hundreds of dollars [Ref. 12:p. 7-14]. The goal of the ALS as directed by law is to drop the cost of placing a pound of payload into LEO to the low hundreds of dollars also [Ref. 24:p. 71]. Perhaps this is an unrealistic goal considering the complexity of the problem.

It appears that the best option for the United States with regard to fulfilling President Bush's challenge is to begin manufacturing a high thrust, hydrocarbon engine, such as the F-1, as the core engine for a new boost vehicle. This hydrocarbon booster could be mated to upper stages with Centaur or other highly reliable, hydrogen-fueled orbital injection engines to deliver payloads to specified orbits with precision and efficiency. This type of booster would be a solution to the short term problem of deploying Freedom by the end of 1999 and it could also provide efficient and reliable lifting capability for the long term as well.

There may still be a need for a more reliable hydrogen fueled STME-like engine in the future. Centaur and other currently available systems may not be able to provide the thrust required for the missions of return to the Moon and manned Martian explorations. Therefore the United States aerospace industry should continue further development of a highly reliable STME-like engine. This engine could then be used as the next generation of upper stage engines for these more long term requirements.

APPENDIX A. ACRONYMS

ABMA	Army Ballistic Missile Agency
ALS	Advanced Launch System
ARPA	Advanced Research Projects Agency
ASRM	Advanced Solid Rocket Booster
DoD	Department of Defense
EOR	Earth Orbit Rendezvous
IOC	Initial Operational Capability
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LOR	Lunar Orbit Rendezvous
LOX	Liquid Oxygen
MSFC	Marshall Space Flight Center
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
RP-1	Hydrocarbon Rocket Propellant
SCC	Space Commerce Corporation
SCE	Shuttle-C Cargo Element
SSME	Space Shuttle Main Engine
STME	Space Transportation Main Engine
STS	Space Transportation System
TFU	Theoretical First Unit
USAF	United States Air Force

LIST OF REFERENCES

1. National Aeronautics and Space Administration, *Report of the 90-Day Study on Human Exploration of the Moon and Mars*, Government Printing Office, Washington, DC, November 1989.

2. National Aeronautics and Space Administration, *America at the Threshold, America's Space Exploration Initiative*, Government Printing Office, Washington, DC, 3 May 1991.

3. Advisory committee on the future of the United States space program, National Aeronautics and Space Administration, *Report of the Advisory Committee on the Future of the United States Space Program,* Government Printing Office, Washington, DC., December 1990.

4. Roger E. Bilstein, *Stages to Saturn, A Technological History of the Apollo/Saturn Launch Vehicles,* Government Printing Office, Washington, DC, 1980.

5. L-Systems Incorporated, *An Evolutionary Approach to a New Independent Launch Vehicle,* A briefing prepared by L-Systems Inc. of El Segundo, California and presented to Lt. General Donald L. Cromer, Commander Air Force Systems Command/Space Systems Division, Revised Edition, 23 October 1990.

6. Robert Truax, *Economics of Satellite Supply,* This article first appeared in the Proceedings of the International Aeronautical Federation 1954. It was sent to the author by Mr. Truax.

7. David Baker, Ph.D., *The Rocket, The History and Development of Rocket and Missile Technology,* Crown Publishers Incorporated, 1978.

8. Paul Coffman, *F-1 Engine,* Hardcopy of the overheads presented in a briefing given at the Marshall Space Flight Center and to the Stafford Commission, November 1990.

9. George P. Sutton, *Rocket Propulsion Elements, an Introduction to the Engineering of Rockets,* Fifth Edition, John Wiley and Sons, 1986.

10. Leland F. Belew, *Skylab, Our First Space Station*, Government Printing Office, Washington, DC, 1977.

11. American Institute of Aeronautics and Astronautics Paper Number 66-837, *Apollo/Saturn V Launch Operations,* by Rocco A. Petrone, 1966.

12. Navy Space Systems Activity, *U. S. Space Launch Systems*, Government Printing Office, Washington, DC, 1977.

13. Telephone conversation between Mr. Paul Coffman, Rocketdyne Division of North American Rockwell and the author, 21 February 1991.

14. Ed Gabris, *Advanced Launch System (ALS) Program Overview and Technical Activities and Budgets*, Hardcopy of the overheads of a briefing prepared by Air Force Systems Command and National Aeronautics and Space Administration, February 1991.

15. Andrew Wilson, *Interavia Space Directory 1989-1990*, Interavia SA, 1989.

16. National Aeronautics and Space Administration, Marshall Space Flight Center, *NASA Fact Sheet, Shuttle-C: Using Existing Systems to Cut Costs and Improve Access to Space,* an undated information document sent to the author by Mr. Bob Lessels, Marshall Space Flight Center Public Affairs Office, February 1991.

17. United Technologies Space Flight Division,*Today's Launch Vehicle for Tomorrow, Shuttle-C,* an undated information document sent to the author by Mr. Bob Lessels, Marshall Space Flight Center Public Affairs Office, February 1991.

18. T. J. Lee, *Space Transportation System for the Future,* Government Printing Office, Washington, DC, 1987.

19. Morton Thiokol Incorporated, Space Operations Division *Expanding the National Space Transportation System*, 1989.

20. National Aeronautics and Space Administration, Marshall Space Flight Center, *Shuttle-C Users Conference Executive Summary, Today's Launch Vehicle for Tomorrow*, Government Printing Office, Washington, DC, 1989.

21. Tom Huntington, "Soviet Booster", *Air and Space Smithsonian*, pp. 90-95, February/March 1991.

22. Telephone conversation between Mr. William B. Wirin, Executive Vice President and Chief Operating Officer Space Commerce Corporation, Houston, Texas, and the author, 11 February 1991.

23. National Space Council, *National Space Launch Strategy*, Executive Office of the President, Washington, DC, 24 July 1991.

24. Congress of the United States, Office of Technology Assessment, *Buyer's Guide, Launch Options for the Future, Special Report,* Government Printing Office, Washington, DC, August, 1988.

25. Telephone conversation between Mr. Al Otto, Stennis Space Center, Advanced Launch System Phase B Studies, and the author, 3 September 1991.

26. James A. Martin, "Comparing Hydrogen and Hydrocarbon Booster Fuels," *American Institute of Aeronautics and Astronautics, Journal of Spacecraft and Rockets, Engineering Notes*, Volume 25, Number 1, pp. 92-94, January/February 1988.

27. D. O. Stanley and T. A. Talay, "Propulsion System Optimization for Advanced Launch System Vehicles," *American Institute of Aeronautics and Astronautics Journal of Spacecraft and Rockets*, Volume 27, Number 3, pp. 246-252, May/June 1990.

28. Telephone conversation between Mr. Tom Irby, Advanced Launch System Program Office, Air Force Systems Command, Space Systems Division, El Segundo, California, and the author, 14 February 1991.

29. Telephone conversation between Mr. Billy Shelton, National Aeronautics and Space Administration, Marshall Space Flight Center, Huntsville, Alabama, and the author, 6 February 1991.

30. Telephone conversation between Mr. Harry Cikanek, Technical Manager, Space Transportation Main Engine, Phase B Studies, National Aeronautics and Space Administration, Marshall Space Flight Center, Huntsville, Alabama, and the author,3 September 1991.

31. Ted Nicholas and Rita Rossi, *U. S. Military Aircraft Data Book, 1991*, Thirteenth Edition, Data Search Associates, October 1990.

32. National Aeronautics and Space Administration, *Saturn V Flight Manual, SA-509*, p.1-6, Government Printing Office, Washington, DC, 15 August 1969, changed 1 January 1971.

33. Interview between Prof. Michael G. Sovereign, Ph.D., Naval Postgraduate School, Monterey, California, and the author, 10 September 1991.

INITIAL DISTRIBUTION LIST

1.	Defense Technical Information Center Cameron Station Alexandria, Virginia 22304-6145	2
2.	Superintendent Attn: Library, Code 0142 Naval Postgraduate School Monterey, California 93943-5002	2
3.	Rudolph Panholzer Space Systems Academic Group, Code SP Naval Postgraduate School Monterey, California 93943-5000	1
4.	Dan C. Boger Administrative Sciences Department, Code AS/Bo Naval Postgraduate School Monterey, California 93943-5000	1
5.	Michael G. Sovereign Operations Research Department, Code OR/So Naval Postgraduate School Monterey, California 93943-5000	1
6.	Robert K. Wolf EJ-32 Marshal Space Flight Center, Alabama 35812	1
7.	Col. James Beall National Space Council Executive Office of the President Washington, DC 20500	1